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AN ANALYSIS OF THERMAL BALANCE IN THE COOLED CABIN OF A SEA KIN--ETC(U)

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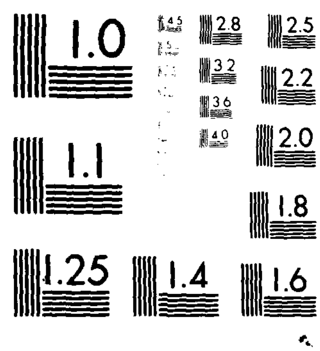

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MECHANICAL ENGINEERING NOTE 378

AN ANALYSIS OF THERMAL BALANCE IN THE COOLED CABIN OF A SEA KING HELICOPTER

by

BRIAN REBBECHI

Approved for Public Release.

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SUMMARY

A thermal balance model has been formulated for cabin cooling of the Sea King Mk. 50 helicopter. The model was evaluated using experimental results obtained during flight trials of a vapour cycle cooling system in a Sea King helicopter of the RAN fleet.

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ABSTRACT

A thermal balance model has been formulated for cabin cooling of the Sea King Mk. 50 helicopter. The model was evaluated using experimental results obtained during flight trials of a vapour cycle cooling system in a Sea King helicopter of the RAN fleet.

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NOMENCLATURE

A	Effective area of fuselage for conductive heat transfer (m^2)
A_e	Area of fuselage in contact with exhaust gas (m^2)
A_o	Surface area (m^2)
A_s	Effective area of fuselage not exposed to solar radiation (m^2)
A_t	Fuselage skin area exposed to solar radiation (m^2)
C_p	Specific heat of air (1·0 kJ/kg)
h_c	Convective heat transfer coefficient (fuselage skin surface to ambient air) (kW/m^2 °C)
H_{tg}	Latent heat of vapourization of water (kJ/kg)
h_o	Overall heat transfer coefficient (kW/m^2 °C)
h_r	Linearised radiation exchange coefficient (kW/m^2 °C)
h_w	Fuselage wall heat transfer coefficient (includes cabin interior air–wall coefficient) (kW/m^2 °C)
m_f	Cabin ventilation air mass flow (kg/s)
m_{fa}	Cooling air mass flow (air-cycle system) (kg/s)
Q_c	Heat removed by cooling unit (kW)
Q_e	Electrical equipment heat load (kW)
Q_{ex}	Heat transfer through fuselage wall in region heated by exhaust gas (kW)
Q_{en}	Heat input to cabin from engine heating and exhaust gas (kW)
Q_f	Fan heating of ventilation air (kW)
Q_{fa}	Heat load of ventilation air (kW)
Q_h	Heating effect (cabin heater) (kW)
Q_L	Condensate cooling load (kW)
Q_m	Metabolic heat output of occupants (kW)
Q_s	Solar heat load transmitted through transparencies (kW)
Q_{si}	Incident solar radiation (kW/m^2)
Q_r	Thermal radiation exchange (kW)
Q_w	Heat conduction through aircraft skin (kW)
Q_{wa}	Heat flow through upper aircraft surface exposed to solar radiation (kW)
Q_{wb}	Heat flow through aircraft surface not exposed to solar radiation (kW)
T_a	Ambient air temperature (°C)
T_c	Cabin mean air temperature (°C)
T_{ex}	Increment above ambient air temperature of air mixed with exhaust gas (°C)

T_{in}	Cabin inlet air temperature (air-cycle cooling system) (°C)
T_{out}	Cabin outlet air temperature (air-cycle cooling system) (°C)
T_r	Radiant temperature of surroundings (°C)
T_s	Surface temperature of an object (°C)
T_{sa}	Surface temperature of aircraft skin exposed to solar radiation (°C)
T_{sb}	Surface temperature of aircraft skin not exposed to solar radiation (°C)
T_{sky}	Effective sky temperature (°C)
δT_{sky}	Difference between ambient and effective sky temperatures (°C)
δ_r	Reduction in humidity ratio (kg moisture/kg dry air)
σ	Stefan–Boltzmann constant ($W/m^2 K^4$)
ϵ_s	Fuselage emissivity at solar radiation wavelengths

1. INTRODUCTION

This study arose from problems of excessive cabin temperatures encountered by the Royal Australian Navy during operation of Sea King Mk. 50 helicopters. These high temperatures were producing adverse effects on crew efficiency, particularly during flights of long duration experienced in anti-submarine missions. The Aeronautical Research Laboratories recorded cabin temperatures and humidities in one Sea King helicopter of the RAN fleet over an extended period; these measurements are reported by Rebbechi and Edwards (1979). A preliminary estimate of the cabin heat loads, together with suggestions for partially alleviating the problem, was made by Rebbechi (1977). The conclusions of this earlier work, however, were that an acceptable cabin environment could only be attained by the use of refrigeration to cool the cabin air.

A vapour cycle cooling unit was subsequently built by ARL for feasibility studies of cabin cooling, and was flight tested in a Sea King helicopter. These flight trials were intended to establish the cooling capacity required in a permanently fitted installation and to evaluate the overall suitability of electrically powered vapour cycle systems. The results of these trials are reported by Rebbechi (1979).

The flight tests were limited, by aircraft availability, to a relatively short testing period in moderately hot outside air temperatures (approximately 30°C). Because of this, some extrapolation from the trials results was necessary to estimate the cooling requirement in climatic extremes; it was in this extrapolation that consideration of the cabin heat balance became necessary.

This study discusses in detail the derivation and evaluation of the heat balance equations for the Sea King, the evaluation being carried out from an analysis of the flight test results of the ARL vapour cycle cooling unit.

Whilst there are a number of published analyses of the heat balance in the cockpit/cabin of fixed-wing aircraft (for example Hughes 1968; Torgeson *et al.* 1955), the author has not sighted similar analyses pertaining specifically to rotary wing aircraft. Analyses have been published of the cockpit/cabin temperatures of uncooled rotary-wing aircraft (Laing 1974; Laing *et al.* 1975). These analyses, however, relate in effect only to the aircraft outer skin temperatures, and do not include heat flows through the aircraft skin. The transient heating and cooling case is considered also, but only by inclusion of an empirical exponential coefficient.

Whilst few helicopters are at present airconditioned, it is to be expected that more will be in the future. For this reason the analysis of cabin heat balance will become increasingly relevant.

2. HEAT BALANCE EQUATION

2.1 Basic Equation of Thermal Balance

The basic equation of thermal balance for the cabin is

$$Q_m + Q_f + Q_{fa} + Q_w + Q_e + Q_{en} + Q_s = Q_c \, * \qquad (1)$$

where Q_m = metabolic heat output of occupants,

Q_f = fan heating of ventilation air,

Q_{fa} = heat load of ventilation air,

Q_w = heat conduction through aircraft skin,

Q_e = equipment heat load,

* The convention is used that in the left-hand side of the equation, heat flowing *into* the cabin is positive.

Q_{en} = heat input to cabin from engine heating and exhaust gas,

Q_s = solar heat load transmitted through transparencies,

Q_c = rate of heat removal by cooling unit.

Each of these parameters (which are in units of kW) is discussed in further detail in Sections 2.1.1 to 2.1.8.

2.1.1 Metabolic Heat Output of Occupants

The metabolic heat generated by an occupant is estimated to be 0·12 kW, for a person engaged in light work, increasing to 0·20 kW when working heavy controls, in rough conditions and during combat sorties (Hughes 1968).

2.1.2 Fan Heating of Ventilation Air

The RAN Sea King is equipped with a ventilation air fan giving an air mass flow of approximately 0·27 kg/s. For tests of the vapour cycle cooling unit, the existing ventilation fan was retained, and the mass flow reduced to 0·045 kg/s by restricting the fan inlet; the temperature rise of air through the fan was then 7°C, resulting in a heat input to the air of 0·32 kW. This method of reducing the flow is appropriate only to a short trial; a more permanent installation would require replacement of the exisiting fan by a more efficient unit.

2.1.3 Heat Load of Ventilation Air

Even without a finite temperature rise in the fan, the ventilation air imposes a heat load on the cabin cooling system because of the temperature differential between the cabin and ambient. The heat load, Q_{fa}, is

$$Q_{fa} = m_f C_p (T_a - T_c) \qquad (C_p \text{ taken as } 1 \cdot 0), \tag{2}$$

where m_f = ventilation air mass flow (for the flight trials equal to 0·045 kg/s),

T_a = ambient air temperature (°C),

T_c = cabin mean air temperature (°C).

2.1.4 Electrical Equipment Heat Load

The equipment heat load arises almost exclusively from heat generated in the cabin avionics. This heat load can range from 0 to 3·11 kW, depending upon the equipment in use. For the cooling unit trials, the electrical load was zero during ground tests, and 1·20 kW during flight tests.

2.1.5 Solar Heat Load Transmitted through Transparencies

Insufficient data was available from the flight trials to determine experimentally solar heating through the transparencies. The heat loading was therefore determined by measuring the projected area of transparencies, which is 2·4 m^2 in plan and 3·0 m^2 if the aircraft is facing to the sun and the altitude of the sun is in the region 45–60°. Assuming, then, a maximum solar radiation intensity of 1·0 kW/m^2, and an average transparency transmittance of 0·83, the heat load, Q_s, then ranges from 1·99 kW to 2·49 kW.

2.1.6 Engine and Exhaust Gas Heating

Heating from the engine and exhaust gas arises from

(*a*) heat conduction from the engine and gearbox compartments, which are above the central cabin area; and (*b*) heating of the aft fuselage skin due to impingement of hot exhaust gas from the engines.

This heating effect is estimated to be constant for a particular flight phase; its magnitude is experimentally derived from the flight test results (see Section 2.2.2).

2.1.7 Heat Conduction Through Aircraft Skin

The heat conduction considered here is that through the fuselage, including transparencies. It was not possible to determine experimentally the *relative* contribution of the various component parts of the fuselage to overall heat conduction. The total area of the cabin and transparencies participating in heat conduction is estimated to be 40 m². This area does not include the floor, which, because of its honeycomb type construction, was not at first thought to participate significantly in heat transfer from the cabin.

The heat flow into (or away from) the cabin, through the fuselage, is considered in two parts:

(*a*) the upper aircraft surface exposed to solar radiation; and

(*b*) the remaining area not exposed to solar radiation.

Then, denoting Q_{wa} as heat flow through part (*a*) above,

$$Q_{wa}+h_r A_t(T_{sa}-T_{sky})+h_c A_t(T_{sa}-T_a)-A_t\epsilon_s Q_{si} = 0 \tag{3}$$

where Q_{wa} = heat flow through upper aircraft surface (kW),

h_r = linearised radiation exchange coefficient* (kW/m² °C),

h_c = convective heat transfer coefficient (fuselage skin surface to ambient air) (kW/m² °C),

A_t = fuselage skin exposed to solar radiation (9 m²),

A_s = effective area of fuselage not exposed to solar radiation (31 m²),

T_a = ambient air temperature (°C),

T_{sa} = aircraft skin temperature (°C) (in region (*a*)),

T_{sky} = effective sky temperature (°C),

Q_{si} = incident solar radiation (kW/m²),

ϵ_s = fuselage emissivity at solar radiation wave-lengths.

Now, as

$$Q_{wa} = A_t h_w(T_{sa}-T_c). \tag{4}$$

where T_c = cabin mean air temperature (°C),

h_w = a heat transfer coefficient which includes the wall thermal conductivity and interior air-wall coefficient (kW/m² °C),

then

$$T_{sa} = (\epsilon_s Q_{si}+T_c h_w+T_{sky}h_r+T_a h_c)/(h_w+h_r+h_c), \tag{5}$$

therefore

$$Q_{wa} = A_t[\epsilon_s Q_{si}+(h_c+h_r)(T_a-T_c)+\delta T_{sky}h_r][h_w/(h_w+h_r+h_c)], \tag{6}$$

where δT_{sky} = differential between ambient and effective sky temperatures

= $(T_{sky}-T_a)$ (°C).

Similarly, for the fuselage area not exposed to solar radiation, and where the effective temperature of the surroundings is taken as equal to ambient temperature,

$$Q_{wb}+A_s(h_r+h_c)(T_{sb}-T_a) = 0, \tag{7}$$

* The linearised radiation exchange coefficient is described in Appendix 1.

where Q_{wb} = heat flow through aircraft surface not exposed to solar radiation (kW),

A_s = fuselage skin area not exposed to solar radiation (m²),

T_{sb} = fuselage skin temperature in region (*b*) (°C).

Then, as

$$Q_{wb} = A_s h_w (T_{sb} - T_c), \tag{8}$$

from Equations (7) and (8),

$$T_{sb} = [T_a(h_r + h_c) + T_c h_w]/(h_w + h_r + h_c) \tag{9}$$

Therefore,

$$Q_{wb} = A_s(h_c + h_r)(T_a - T_c)[h_w/(h_w + h_r + h_c)] \tag{10}$$

Thus, the total heat flow, Q_w, is given by

$$Q_w = Q_{wa} + Q_{wb}, \tag{11}$$

$$= [A_0(h_c + h_r)(T_a - T_c) + A_t(\epsilon_s Q_{si} + \delta T_{sky} h_r)][h_w/(h_w + h_r + h_c)], \tag{12}$$

where $A_0 = A_t + A_s$. (13)

It will be noted from the foregoing equations (3) to (9), that a uniform fuselage conductivity, h_w, is assumed, although the skin temperature reached in the two regions (*a*) and (*b*), as given by Equation (5) for region (*a*), and Equation (9) for region (*b*), is considered in general to differ.

2.1.8 Heat Removed by Cooling Unit

The heat removed by the cooling unit is ducted directly overboard in the condenser cooling air. The net sensible heat removed is found from a heat balance of air passing through the cooling unit evaporator.

2.2 Experimental Derivation of Unknown Factors in Thermal Balance Equation

2.2.1 Trials Results

The experimental programme was limited by considerations of aircraft serviceability, to the extent that useful results are available only from four ground tests (each of approximately two hours duration to enable steady-state conditions to be reached), and one flight test, of three hours duration. Full details of the flight trials results are given by Rebbechi (1979); in this report (Figs. 1-3) the results are summarised by presentation of the cabin temperatures, outside air temperatures and cooling effects for two ground tests and one flight test.

2.2.2 Fuselage Conductivity

The fuselage conductivity is evaluated from the two ground cooling tests (Figs. 1 and 2), where the aicraft was parked in full solar radiation. The parameters relating to the two tests are listed in Table 1, substitution of these parameters into Equation (12) results in two equations, the two unknowns being h_w and $\epsilon_s Q_{si}$. Solution of these two equations gives the result

$$h_w = 0{\cdot}0124 \quad \text{(kW/m}^2\text{ C)}, \tag{14}$$

$$\epsilon_s Q_{si} = 0{\cdot}39 \quad \text{(kW/m}^2\text{)}. \tag{15}$$

Hence, at the (assumed) value of 1·0 kW/m² for Q_{si}, $\epsilon_s = 0{\cdot}39$.

The relative heat transmitted, and skin temperatures for each of the two regions (*a*) and (*b*) can be calculated from Equation (5) and (9). The results are summarised in Table 2.

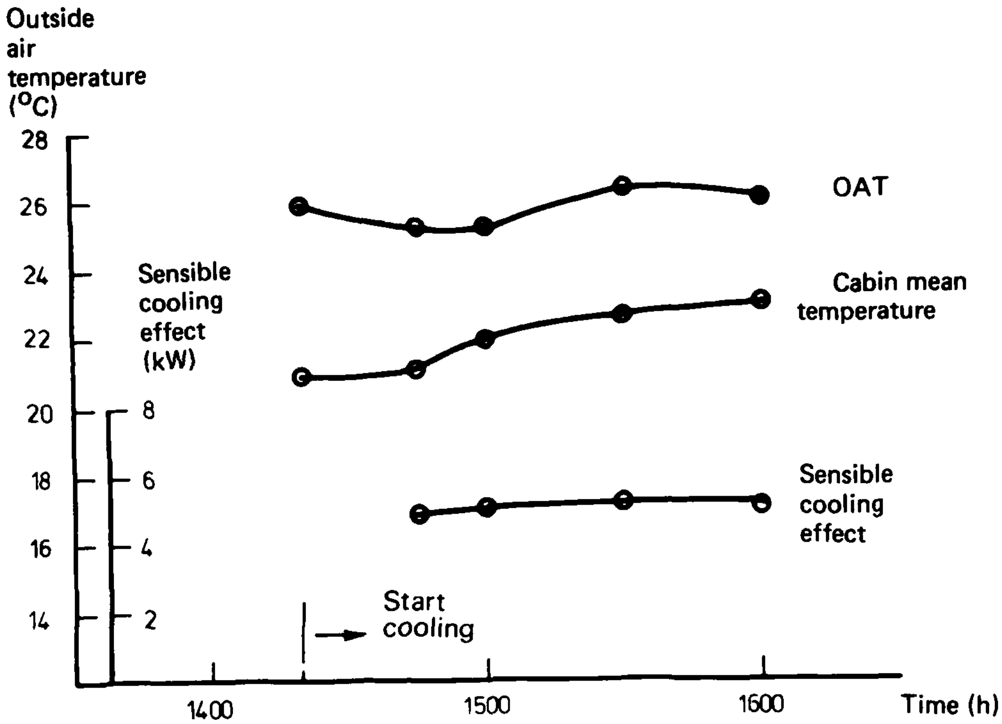

FIG 1 TEMPERATURES AND COOLING EFFECT FOR GROUND RUN IN SOLAR RADIATION FEB. 7 1979.

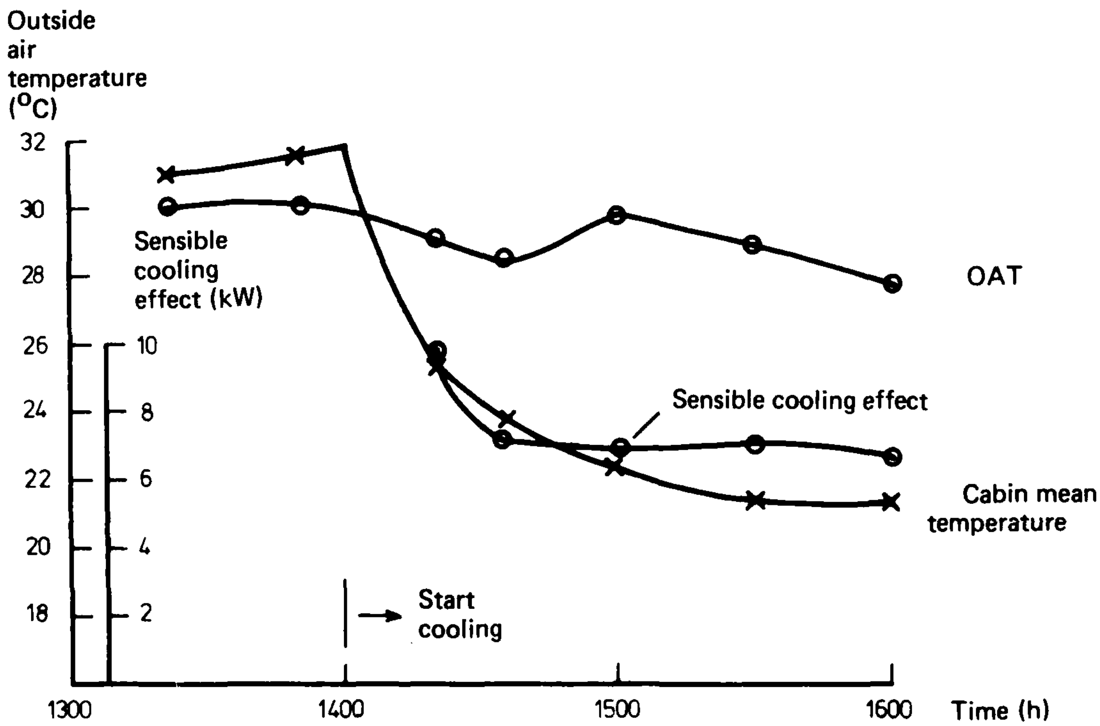

FIG 2 TEMPERATURES AND COOLING EFFECT FOR GROUND RUN FEB. 1 1979.

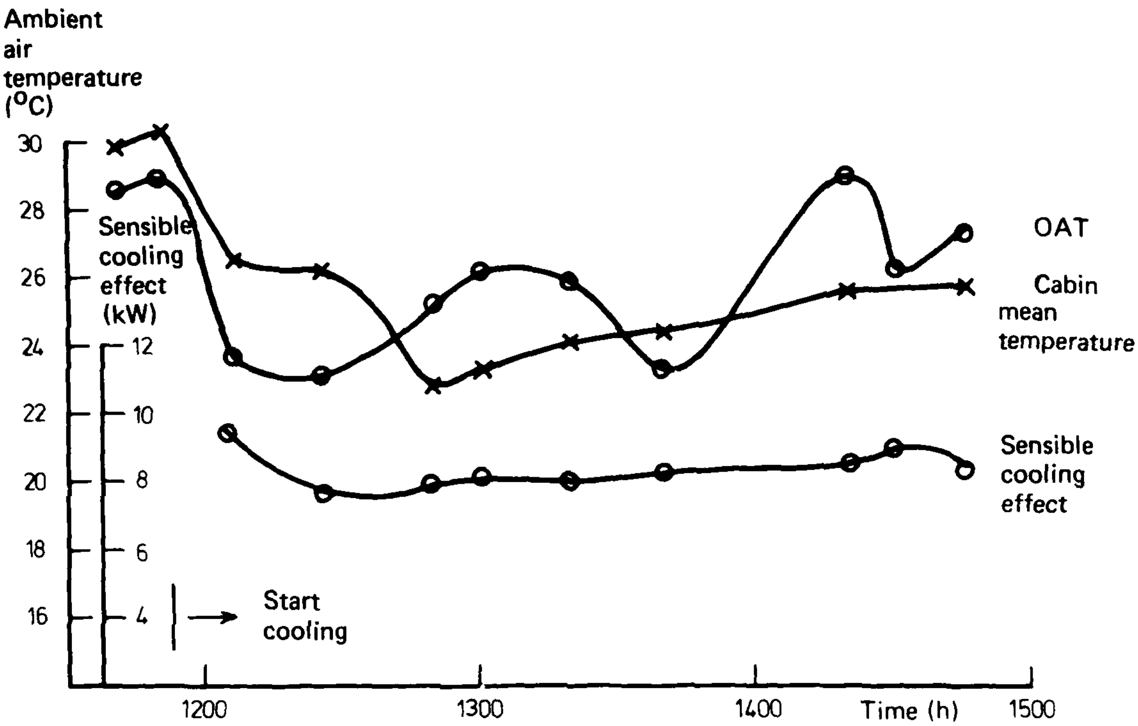

FIG 3 TEMPERATURES AND COOLING EFFECT FOR FLIGHT TEST FEB. 2 1979.

TABLE 1

Test Parameters

Parameters	Test 1 (7 February)	Test 2 (1 February)
T_a (Figs. 1 and 2) (°C)	26·0	28·4
T_c (Figs. 1 and 2) (°C)	23·0	21·3
$Q_{fa} = 0{\cdot}045(T_a - T_c)$ (kW)	0·14	0·32
Q_f (kW)	0·32	0·32
Q_m (kW)	0·12	0·12
Q_s (kW)	2·49	2·49
Q_c (Figs. 1 and 2) (kW)	5·11	6·63
A_e (m²)	9·0	9·0
A_s (m²)	31·0	31·0
h_r (Appendix 1) (kW/m² °C)	0·0056	0·0056
h_c (Fig. 4) (kW/m² °C)	0·015 (3 kts)	0·034 (14 kts)
Q_w ($Q_w = Q_c - Q_s - Q_m - Q_{fa}$) (kW)	2·04	3·38
δT_{sky} (Duffie and Beckman 1974) (°C)	−12·0	−12·0

TABLE 2

Calculated Results from Tests 1 and 2

Parameter	Test 1	Test 2
T_a (°C)	26·0	28·4
T_c (°C)	23·0	21·3
Q_{wa} (kW)	1·32	1·31
Q_{wb} (kW)	0·72	2·07
T_{sa} (°C)	34·9	33·0
T_{sb} (°C)	24·9	26·7

2.2.3 Engine and Exhaust Gas Heating

Heating from this source is caused by direct conduction from the engine and gearbox compartments, and heating of the aft part of the fuselage by engine exhaust gas. It is taken to be a constant, as it is assumed that the mixed flow of exhaust gas and ambient air which impinges on the fuselage is at a constant temperature differential above ambient, hence

$$Q_{ex} = h_o(T_a + T_{ex} - T_c)A_e \qquad (16)$$

and

$$Q_{ex} = A_e h_o T_{ex} + h_o(T_a - T_c)A_e \qquad (17)$$

where Q_{ex} = heat transfer through fuselage wall in region heated by exhaust gas (kW),

A_e = area of fuselage in contact with exhaust gas (m²),

h_o = overall heat transfer coefficient (kW/m² °C),

T_{ex} = increment above ambient air temperature of air mixed with exhaust gas (°C).

Thus, as $A_e h_o T_{ex}$ is a constant for a particular flight phase, and the term $h_o(T_a - T_c)A_e$ is included in Q_w (see Section 2.1.7) then Q_{ex} will be constant. A change in A_e could be expected in the hover phase compared with forward flight—an effect which has been observed by Rebbechi and Edwards (1979) in their measurements of fuselage skin temperature. The parameter T_{ex} can also be expected to change with engine power settings. Direct conduction from the engine compartment to the cabin interior will be assumed constant.

To evaluate Q_{en} the results of the flight test (Fig. 3) are used. These are summarised as follows:

T_a	27·6°C
T_c	25·6°C
Q_c	8·20 kW
Q_m	0·72 kW
Q_e	1·20 kW
Q_s	2·49 kW
Q_f	0·32 kW
Q_{fa}	$0{\cdot}045(T_a - T_c) = 0{\cdot}09$ kW
Aircraft velocity	100 kn (51 m/s)

Before substituting these results into the basic equation of thermal balance—Equation (1)—considerable simplifications can be made to the parameter Q_w (Equation (12)). At an aircraft speed of 100 kn (51 m/s), the convective heat transfer coefficient is much greater than either of the coefficients h_r, h_w (see Fig. 4). Then, Equation (12) can be reduced to the form

$$Q_w = Ah_w(T_a - T_c) \tag{18}$$

This simplification is equivalent to putting $T_s = T_a$.*

From Equation (1) then, substituting the flight test parameters gives the result

$$Q_{en} = 2{\cdot}39 \text{ kW} \tag{19}$$

2.3 Summary of Thermal Balance Equation

2.3.1 Parked Aircraft

For the parked aircraft case the heat balance equation, (1), can now be summarised as

$$Q_m + Q_e + Q_s + Q_{fa} + Q_w = Q_c \tag{20}$$

This assumes that engine and exhaust heating are absent, and that, by the use of a more efficient ventilation fan, Q_f is reduced to zero. From Equation (12),

$$Q_w = [A(h_c + h_r)(T_a - T_c) + A_t(\epsilon_s Q_{si} + \delta T_{sky} h_r)][h_w/(h_w + h_r + h_c)] \tag{21}$$

Values of the parameters in Equations (20) and (21) are as follows:

Q_m	0·12 kW per person
Q_e	0–3·11 kW, depending on equipment in use
Q_s	0–2·49 kW, depending on intensity of solar radiation
Q_{fa}	$C_p m_f(T_a - T_c)$ kW, where m_f for these trials is 0·045 kg/s
A	40 m^2

* The skin temperature (T_s) will actually closely approach the stagnation temperature—while this effect is very significant for high speed aircraft ($M > 0{\cdot}5$), for a helicopter speed of 100 kn the stagnation temperature rise above ambient is very small (1·2°C) and thus there is little point in introducing this additional factor for such low speeds.

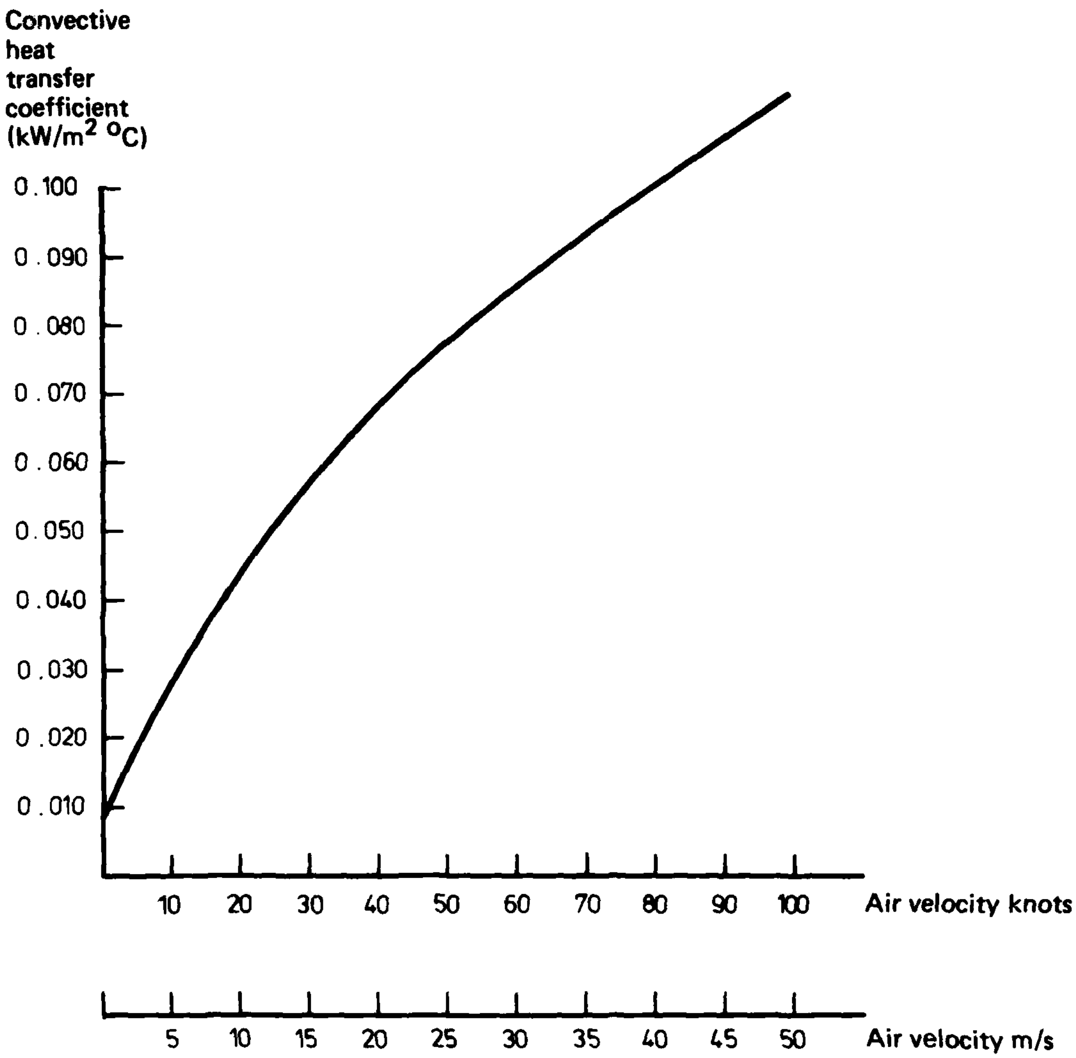

FIG 4 CONVECTIVE HEAT TRANSFER COEFFICIENT VS. AIR VELOCITY
(from Torgeson et al (1955), and Barned and O'Brien (1970)).

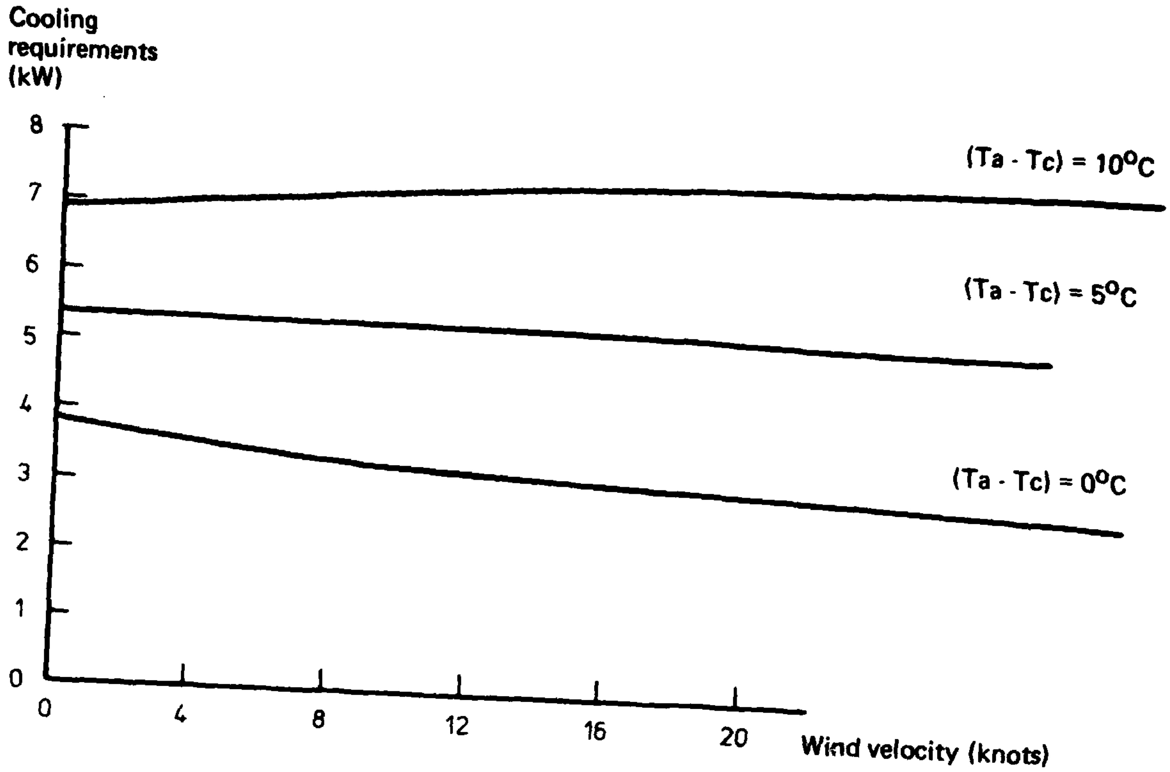

FIG 5 GROUND COOLING REQUIREMENTS (DOORS CLOSED, ENGINES NOT OPERATING, $Q_e = 0$)

A_t	9 m^2
ϵ_s	0·39
Q_{st}	0–1·0 kW/m^2, depending on solar radiation
δT_{sky}	-12°C for clear sky (Duffie and Beckman 1974)
h_c	See Figure 4—a function of ground wind velocity (kW/m^2 °C)
h_r	0·0056 kW/m^2 °C
h_w	0·0124 kW/m^2 °C

The parked aircraft cooling requirements, as given by Equation (20), for several cabin temperature differentials $(T_a - T_c)$ are plotted against wind velocity in Figure 5. This figure is plotted for the case of full solar radiation ($Q_s = 2{\cdot}49$), no electrical equipment load ($Q_e = 0$), normal ventilation, doors closed, but no occupants.

2.3.2 Cooling of Aircraft Cabin in Flight

The heat balance equation can be summarised as

$$Q_m + Q_e + Q_s + 2{\cdot}39 + (T_a - T_c)(Ah_w + C_p m_f) = Q_c, \tag{22}$$

where $Q_m = 0{\cdot}12$ kW per person

$Q_e = 0{\cdot}5$–$3{\cdot}11$ kW, depending on whether sonar and/or radar in use

$Q_s = 0$–$2{\cdot}49$ kW, depending upon level of solar radiation

$A = 40$ m^2

$h_w = 0{\cdot}0124$ kW/m^2 °C

$m_f = 0{\cdot}045$ kg/s, for trials configuration

$C_p = 1{\cdot}0$ kJ/kg °C

Then, for a crew of four and full solar radiation, Equation (22) becomes

$$Q_e + 5{\cdot}36 + 0{\cdot}54(T_a - T_c) = Q_c$$

This equation is presented graphically in Figure 6, where the cabin differential $(T_a - T_c)$ is plotted versus cooling requirement Q_c, for several values of electrical equipment heat load (Q_e).

2.3.3 Heating of Aircraft Cabin in Flight

From the heat balance equation, (22), for in-flight cooling, and putting $Q_h = -Q_c$, where Q_h = heating effect (kW), the heat balance equation for in-flight heating becomes

$$Q_h = (T_c - T_a)(Ah_w + m_f) - Q_m - Q_e - Q_s - 2{\cdot}39 \tag{24}$$

The worst case for heating will be when $Q_s = 0$ (very heavily overcast, or night), $Q_m = 0{\cdot}48$ (minimum crew), and $Q_e = 0{\cdot}5$ (minimum avionics). Equation (24) then becomes, substituting appropriate values for A and h_w,

$$Q_h = (T_c - T_a)(0{\cdot}50 + m_f) - 3{\cdot}37 \tag{25}$$

2.4. Discussion

The previous analysis of the heat balance for a Sea King helicopter by the author (Rebbechi 1977) considerably underestimated both the cabin wall heat transfer coefficient (h_w), and the heating due to engine and exhaust gas (Q_{en}). It appears, then, that the fuselage insulation is much less effective than previously thought, and that substantial "leakage" of heat is occurring through bulkheads and airframe fittings, in a similar manner to that described by Hughes (1968)

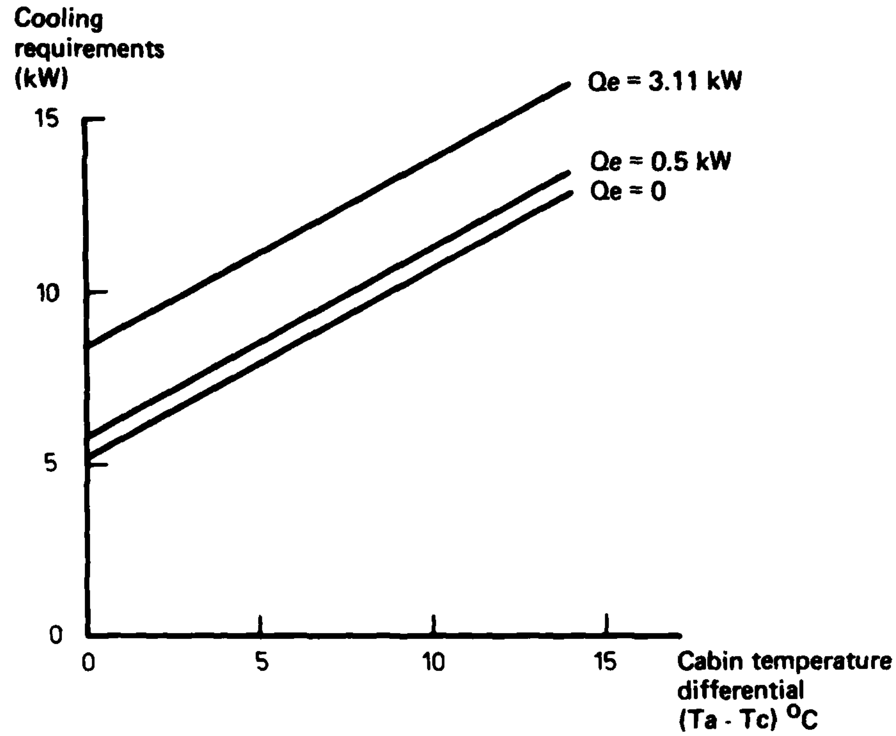

FIG 6 SENSIBLE COOLING REQUIREMENTS FOR IN-FLIGHT AIRCRAFT

Engine exhaust
(port side)

FIG 7 AIRCRAFT ENGINE EXHAUST OUTLETS.

for fixed-wing aircraft (although the Sea King interior air velocities are considerably lower). Even so, the heat transfer coefficient h_w evaluated here as being 0·0124 kW/m² °C is close to the value to be expected for an uninsulated cabin; it is possible, then, that the floor contributes to heat transfer from the cabin, though this was initially discounted.

The experimental technique used here to estimate h_w is not ideal, as it was based upon ground tests. It would have been preferable to evaluate h_w from a range of flight tests, in which parameters such as Q_s and Q_e were constant, and where Q_c, and hence T_c, varied over as wide a range as possible. However, the flight testing was very limited, and had to be eventually discontinued because of problems with aircraft serviceability (not connected with the airconditioning unit installation).

Several tests were carried out with the aircraft parked in the hangar in still-air conditions; however, to utilise these results an accurate estimate of the effective temperature of the surroundings is necessary, as in still air the convective heat transfer coefficient is approximately equal to the linearised radiation exchange coefficient.

The quite large heat input (2·39 kW) due to heating from the engine and exhaust gas is not surprising in view of the high aft fuselage skin temperatures; Rebbechi and Edwards (1979) report skin temperatures in this region up to 20 °C above ambient. These high temperatures would be due in part to the exhaust outlets being flush with the engine cowling, as can be seen from Figure 7. It is also likely that the heat input from this source would vary with aircraft flight phase and engine power settings, as discussed in Section 2.2.2.

3. APPLICATION OF HEAT BALANCE EQUATION TO VARIOUS TYPES OF COOLING SYSTEMS

3.1 Vapour Cycle Cooling Systems

A vapour cycle cooling system will typically have a large recirculating air mass flow component, and a small throughflow component (for ventilation purposes only). Pressurization requirements are generally absent for helicopter operations. The heat balance equation (22),

$$Q_m + Q_e + Q_s + 2{\cdot}39 + (T_a - T_c)(Ah_w + C_p m_f) = Q_c \tag{22}$$

is for sensible heat flows only. As the evaporator temperature is often less than the dew point of the air passing through the evaporator, condensation of moisture in the air will occur, giving rise to an additional cooling load

$$Q_L = m_f H_{fg} \delta r \tag{26}$$

where Q_L = condensate cooling load (kW)

m_f = cabin ventilation air mass flow (kg/s)

H_{fg} = latent heat of vapourisation of water = 2454 kJ/kg

δr = reduction in humidity ratio (kg moisture/kg dry air)

Equation (26) is for steady-state conditions; a larger initial condensate load will occur on start-up, until the moisture content of all of the cabin air is reduced to the steady-state value.

The total cooling load Q_T is then

$$Q_T = Q_m + Q_e + Q_s + 2{\cdot}39 + Ah_w(T_a - T_c) + m_f(T_a - T_c + H_{fg}\delta r). \tag{27}$$

An interesting feature of Equation (27) is that the effect on the cooling load of ambient air leaking in through window and door seals arises predominantly from the effect on the condensate cooling load, rather than the effect on the sensible cooling load. The total heat load due to ventilation air is given by the last term in Equation (27), that is $m_f(T_a - T_c + H_{fg}\delta r)$ (taking C_p as unity). In extreme climatic conditions, typical values of $H_{fg}\delta r$ and $(T_a - T_c)$ are 29·5 (δr = 0·012) and 10 °C respectively, resulting then in a condensate cooling load three times that of the sensible cooling load. For a ventilation airflow of 0·045 kg/s, as used in the Sea King cooling unit trials, the heat loads arising from the ventilation air would then be 0·45 kW (sensible) and 1·32 kW (condensate). The importance of reducing extraneous ventilation airflows is therefore clear.

3.2 Air Cycle Cooling Systems

An air-cycle cooling system will either introduce a supply of cooling air to the cabin (typically at a temperature just above 0°C to avoid freezing problems), or, as in recent versions, introduce also a recirculation component. Considering here the simplest case of a steady supply of cooling air, the cabin cooling effect Q_c is

$$Q_c = m_{ta} C_p (T_{out} - T_{in}) \tag{28}$$

where m_{ta} = cooling air mass flow (kg/s),

T_{out} = cabin outlet air temperature (°C)

T_{in} = cabin inlet air temperature (°C)

C_p = specific heat of air (1·0 kJ/kg °C)

Where the cabin volume is large compared with the cooling airflow, the mean cabin air temperature (T_c) will approximate to the outlet air temperature. Then, combining Equations (22) and (28) to eliminate Q_c,

$$T_c = (Q_m + Q_e + Q_s + 2 \cdot 39 + T_a A h_w + m_{ta} T_{in}) / (m_{ta} + A h_w) \tag{29}$$

4. RECOMMENDATIONS FOR FURTHER WORK

Whilst this analysis of the Sea King heat balance has fulfilled its primary objective, that is to enable the specification of the cooling requirements for a prototype production installation, several questions remain unanswered. These questions relate to the reasons for the high fuselage wall heat transfer coefficient h_w, and the degree to which the exhaust gas heating may vary with flight phase and engine power. Answers to these questions would improve the confidence of predictions of cabin heat balance for other helicopter types.

The use of an infra-red (thermal imaging) camera would greatly facilitate this work, as more complete mapping of skin temperature could then be made, both on the ground and in flight (from another accompanying helicopter). Perhaps more importantly, *differences* in skin temperature show up very readily with this type of instrumentation, thus enabling recognition of the effect that bulkheads, etc., may be having upon heat transfer, identification of the areas affected by exhaust gas heating, and also the source of air leakages (which show up as a localised change in temperature of the structure). Assessment of the effect that bulkheads, etc., have on heat transfer and overall thermal conductivities could be obtained quite accurately using electric heaters in the aircraft, rather than by the more difficult method of using cabin cooling. This would only be applicable for ground running, but if carried out in conjunction with thermal imaging, could yield very useful results.

5. CONCLUSIONS

This note has presented a detailed analysis of the heat balance of a cooled helicopter, both in flight and parked. Several significant features emerging from this analysis are:

(1) Considerable differences exist between the heat balance for an aircraft when parked, and in flight.

(2) Significant heat inputs to the cabin are caused by exhaust gas heating of the skin, and conduction from the engine area; for the Sea King the heat input attributable to these sources was 2·39 kW.

(3) Where a vapour cycle cooling system is used the cooling load attributable to condensation of moisture in air entering the cabin via the ventilation system and through inadvertent leakage can be three times that required to extract sensible heat from this air.

Thus leakage of outside air into the cabin, for example from around ill-fitting doors and windows, can cause a considerable increase in overall cooling requirements. Where an air-cycle cooling system is used, the effect of this leakage is much less, as only the sensible heat load of the leakage air is relevant to this case.

(4) While the heat balance equations derived herein may be used with reasonable confidence for the present aircraft, several anomalous features, for instance conduction through the (nominally insulated) cabin walls, would prevent accurate extrapolation to other types of helicopter. Further experimental investigations, using more accurate and detailed measurements of skin temperatures, would be required before a more generalised model of cabin heat transfer could be derived.

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APPENDIX 1

Linearised Thermal Radiation Exchange Coefficient

The rate of thermal radiation exchange, Q_r, between a body at temperature T_s (°C) and its surroundings at temperature T_r (°C) is given by the rigorous expression:

$$Q_r = A\epsilon_r\sigma[(T_r+273)^4-(T_s+273)^4] \quad \text{(A1)}$$

where A = surface area of the body (m^2)

ϵ_r = emissivity of the body at temperature T*

σ = Stefan-Boltzmann constant ($5 \cdot 67 \times 10^{-8}$ W/m^2 K^4)

However, for painted aircraft surfaces, at near-ambient temperatures, radiating to the sky or to the ground, ϵ_r is close to unity, and as the difference between T_r and T_s in the cases considered rarely exceeds 50°C, Expression (A1) can be simplified to

$$Q_r = 0 \cdot 0056(T_r - T_s)A, \quad \text{(kW)} \quad \text{(A2)}$$

$$= h_r(T_r - T_s)A$$

hence

$$h_r = 0 \cdot 0056 \text{ (kW/m}^2\text{ °C)}$$

Note that Q_r is positive (heat flowing into the body) when $T_r > T_s$.

* The effecti.e .nissivity is a function of the emissivities of the radiating body and the surroundings, and of the shape of the body.

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